MSC-04112
SUPPLEMENT 4

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

APOLLO 14 MISSION REPORT

SUPPLEMENT 4

ASCENT PROPULSION SYSTEM FINAL FLIGHT EVALUATION

MANNED SPACECRAFT CENTER
HOUSTON, TEXAS
MAY 1972

PROJECT TECHNICAL REPORT

APOLLO 14
LM-8
ASCENT PROPULSION SYSTEM
FINAL FLIGHT EVALUATION

Follow-on Contract to NAS 9-8166 SEPTEMBER 1971

Prepared for
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
MANNED SPACECRAFT CENTER
HOUSTON, TEXAS

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The purpose of this report is to present the results of the postflight analysis of the Ascent Propulsion System (APS) performance during the Apollo 14 Mission. It is a supplement to the Apollo 14 Mission Report. Determination of the APS steady-state performance under actual flight environmental conditions was the primary objective of the analysis. Included in the report are such information as is required to provide a comprehensive description of APS performance during the Apollo 14 Mission.

Major additions and changes to results as presented in the mission report (Reference 1) are listed below:

1) Calculated performance values for the APS Lunar Liftoff burn.
2) Discussion of analysis techniques, problems and assumptions.
3) Comparison of postflight analysis and preflight prediction.
4) Reaction Control Systems (RCS) duty cycle included in APS Performance analysis.
5) Transient performance analysis.
6) Revised estimates of propellant consumption.

2. SUMMARY

The duty cycle for the LM-8 APS consisted of two firings, an ascent stage liftoff from the lunar surface and the Terminal Phase Initiation (TPI) burn. APS performance for the first firing was evaluated and found to be satisfactory. No propulsion data were received from the second APS burn; however, all indications were that the burn was nominal.

Engine ignition for the APS lunar liftoff burn occurred at an Apollo elapsed time (AET) of 141:45:40.0 (hours:minutes:seconds). Total burn duration was 432.1 seconds.

Average steady-state engine performance parameters for the burn are as follows:

Thrust - 3461 lbf

Isp - 309.7 sec

Mixture Ratio - 1.598

All performance parameters were well within their expected 3-sigma limits. Calculated engine throat erosion at engine cutoff for LM-8 APS was approximately 2 percent greater than predicted.

The Apollo 14 Mission was the seventh flight, and the sixth manned flight, of the Lunar Module (LM). The mission accomplished the third successful lunar landing.

Launch from Kennedy Space Center (KSC) occurred at 4:03 p.m. Eastern Standard Time (EST) on 31 January 1971. The launch phase was normal with the exception of a delay ($\sim$ 40 minutes) due to weather. Following earth orbit insertion, the S-IVB stage was restarted and performed the Translunar Injection (TLI) maneuver at approximately 2-1/2 hours Apollo Elapsed Time (AET). CSM-LM docking occurred at approximately 5 hours AET. Separation of the docked vehicles from the S-IVB was accomplished 50 minutes later. Two midcourse correction burns were performed by the Service Propulsion System (SPS) during the translunar phase of the mission. The Lunar Orbit Insertion (LOI-1) and Descent Orbit Insertion (DOI) maneuvers were also performed using the SPS. The LOI-1 burn was conducted at approximately 82 hours AET and the DOI burn occurred approximately 4 hours later. The Descent Propulsion Systems (DPS) duty cycle consisted of one firing; the Powered Descent Initiation (PDI) burn. Engine ignition time for the PDI burn was approximately 108 hours AET. Lunar Landing occurred at 108:15:09 (hours:minutes:seconds) AET. Ascent Propulsion System (APS) ignition time for lunar liftoff was 141:45:40.0 AET with engine cutoff being commanded at 141:52:52.1 AET for an APS burn duration of 432.1 seconds. The 4-second terminal phase initiation (TPI) maneuver began at 142:30:51.1 and was performed for the first time by the APS (previously performed by the LM RCS). CSM-LM docking was accomplished at approximately 143-1/2 hours AET. After crew and equipment transfer had been effected, the LM was jettisoned. Exact data concerning ascent stage main engine ignition and cutoff times and the associated velocity changes are shown in Table 1.

After a separation maneuver using the SM RCS, the LM was maneuvered with its RCS so as to impact on the lunar surface. Lunar impact occurred at approximately 147-3/4 hours AET, terminating APS telemetry data.

The Apollo 14 LM-8 APS was equipped with Rocketdyne Engine S/N 0006C. APS engine performance characterization equations used in preflight analyses and as a basis for the postflight analysis are found in Reference 2. Engine acceptance test data used in the determination of performance are from Reference 3. Physical characteristics of the engine and feed system are presented in Table 2.

Analysis Technique

Determination of APS steady-state performance during the lunar orbit insertion burn was the primary objective of the LM-8 postflight analysis. The insertion burn duration was 432.1 seconds, engine on to engine off command. In addition to the orbital insertion maneuver the APS was used to perform the Terminal Phase Initiation (TPI) burn. Burn duration for TPI was approximately 3.6 seconds. No propulsion system telemetry data are available from the TPI burn since the spacecraft was behind the moon.

The APS postflight analysis was conducted using the Apollo Propulsion Analysis Program (PAP) as the primary computational tool. Additionally, the Ascent Propulsion Subsystem Mixture Ratio Program (MRAPS) was used in an iterative technique with PAP to determine the vehicle propellant mixture ratio. PAP utilizes a minimum variance technique to establish the "best" correlation between an engine characterization model, derived from ground test data, and selected flight measurements. The program embodies error models for the various flight and ground test data that are used as program inputs and combines these with the empirically derived engine characterization equations. Successive iterations through the program result in estimations of system performance history and weights which "best," in a minimum variance sense, reconcile the available data. The MRAPS Program is based on the fact that as APS propellants are consumed the X and Y coordinates of the vehicle center of gravity (C.G.) shift. The movement of the vehicle C.G. results in a change in the torque about the Z-axis from the APS engine thrust. This torque is balanced by the RCS thrusters. The Y C.G. is located by solving the APS-RCS torque balance equation and the X C.G. is computed from the X C.G. of the inert vehicle and the propellant remaining. The slope of the line plotting the position of the Y C.G. is used to determine an estimate of APS mixture ratio and an iterative technique is used to converge on a final value of APS mixture ratio. Reference 4 presents a more detailed explanation of the operation of the MRAPS program and the underlying theory which it implements.

An initial estimate of the ascent stage weight at lunar liftoff of 10780 lbm was obtained from Reference 5. Ascent stage damp weight (total spacecraft weight less APS propellants) was considered to be constant throughout the run, except for a 0.03 lbm/sec overboard flowrate which accounts for ablative nozzle erosion.

RCS propellant usage and thrust histories were obtained from an analysis of the RCS bi-level measurements. All RCS consumption during the ascent burn was from the APS tanks. Table 3 presents a summary of propellant usage, including RCS consumption, from the APS tanks during the ascent burn. Propellant densities used in the program were based on equations from Reference 6, adjusted by measured density data for the LM-8 flight given in the Spacecraft Operational Data Book (SODB), Reference 7. Oxidizer and fuel temperatures were taken from flight measurement data and were 69.8°F and 69.5°F, respectively. These temperatures were considered to be constant throughout the segment of burn analyzed. The following flight measurement data were used in the analysis of the LM-8 APS burn: engine chamber pressure, engine interface pressures, vehicle thrust acceleration, propellant tank bulk temperatures, helium regulator outlet pressures, engine on-off commands, helium tank pressure measurements, and RCS thruster solenoid bi-level measurements. Measurement numbers and other data pertinent to the above measurements, with the exception of RCS bi-levels, in the appendix to this report.

Flight Data Analysis and Results

A 390-second segment of the APS lunar liftoff burn was selected to be analyzed for the purpose of determining steady-state performance. The segment of the burn analyzed begins at 141:46:00.0 AET, 20.0 seconds after ignition, and ends at 141.52:30.0 AET, 22.1 seconds prior to cutoff. The periods immediately following ignition and immediately prior to engine cutoff are not included in order to minimize any errors resulting from data filtering spans which included the start and shutdown transients. APS engine propellant consumption during the burn is presented in Table 3. Propellant consumption from engine on command to the start of the steady-state analysis segment and from the end of the steady-state analysis to the beginning of chamber pressure decay was extrapolated from steady-state analysis results.

The primary engine performance determinations made during the LM-8 postflight analysis are as follow. All average values are over the 390-second period of steady-state analysis.

1) Average APS specific impulse was 309.7 seconds.
2) Average APS mixture ratio was determined to be 1.598
3) Average APS thrust was 3461 lbf.
4) Engine throat erosion was 2% greater than predicted at 400 seconds from ignition.

An extrapolation of the APS steady-state analysis to include the entire burn, with the exception of ignition and shutdown transients, resulted in an average specific impulse, thrust, and mixture ratio of the same value as the 390 second burn segment. LM-8 APS performance was slightly less than predicted with average engine specific impulse being less than the predicted average value by 0.6 second.

The general solution approach used in the LM-8 flight evaluation was to calculate a vehicle weight (including propellant loads) for the beginning of the segment of burn used to analyze steady-state performance and then allow the Apollo Propulsion Analysis Program to vary this weight and other selected performance parameters (state variables) in order to achieve an acceptable data match. The PAP simulations were made using the previously discussed APS engine characterization model driven by engine interface pressures. Raw flight interface pressure measurement data were first filtered with a sliding arc filter and then, because of excessive distortion, these data were further smoothed using a fifth degree curve fit.

Simulation of RCS activity was accomplished by calculating individual thruster "on" time from the RCS accumulated "on" time data and using this to determine an impulse imparted to the vehicle in the direction of the APS engine thrust vector. This impulse was then converted to an effective thrust over a discrete time interval (10 seconds). RCS propellant flowrates for the same intervals were calculated as a percentage of a nominal consumption of 0.36 lbm/sec. The percentage of nominal consumption is equivalent to the value of the effective thrust as a percentage of a 100 lbf nominal thrust. RCS propellant consumption was verified by comparing the integrated value obtained from the method described above with the total consumption determined by multiplying total system "on" time by the nominal 0.36 lbm/sec flowrate. A small adjustment was made to propellant mass overboard to account for consumption of RCS engines in a plane perpendicular to the thrust vector of the APS engine. The resulting thrust and flowrate data were characterized with fifth degree curve fits, as functions of time, general give the calculated instantaneous thrust and flowrates for the RCS thrusters due to the method of calculation and variations in thrust levels for varying engine pulse durations, but over the total time period evaluated they will satisfactorily approximate the total impulse and mass change. At discrete time points when the RCS residuals (curve fit minus calculated data) were excessive, minor adjustments were made to the RCS thrust and flowrate curve fits to reduce the residuals.

Initial PAP simulation results based on the input data outlined above were not acceptable in that the residuals (differences between the filtered flight data and the program calculated values) indicated time correlated errors. The acceleration residuals had a positive slope indicating that an increase in calculated acceleration with flight time was required to minimize the residual error. This effect may be gained by increasing engine flowrates and/or increasing engine thrust on a time basis. The chamber pressure residuals had a negative slope which, in combination with the need for an increase in calculated acceleration, indicated that a greater than predicted throat erosion rate was necessary. A revised throat erosion curve was calculated using the partial derivatives of throat area with respect to acceleration at ten-second intervals throughout the run. The revision of the throat area curve included increasing the initial value to 16.45 in^2, about 0.7 percent larger than the preflight value. The inclusion of this calculated throat area curve in the analysis program resulted in an excellent acceleration match with a near zero mean and no significant slope. The derived throat erosion curve was 2 percent greater than predicted at approximately 400 seconds after ignition. Figure 1 shows the calculated throat area curve in comparison with the predicted curve for LM-8.

The chamber pressure match resulting from the inclusion of the calculated throat area curve was not as good as might have been expected. The resulting residual sloped upward for approximately the first 260 seconds of the burn and then leveled off for the remainder of the burn. However, the shape error in the residual was one that has been in evidence in past APS analyses; i.e., the residual curve sloped upward for approximately 200 seconds following ignition then leveled off for the remainder of the burn. It has been hypothesized that this error is the result of a thermally induced drift on the chamber pressure measurement. The drift for LM-8 was assumed to be .0017 psi/sec for the first 260 seconds of the burn and approximately zero thereafter. This adjustment was applied to the calculated value of the chamber pressure measurement. Additionally, the chamber pressure measurement was determined to be biased by -.4[1] psi based on measurement data prior to ignition. The residual match seen in Figure 3 incorporates both the bias and the drift mentioned above.

After an acceptable PAP simulation had been made the results of the MRAPS program were incorporated. PAP made changes in the interface pressure biases of -.1 psi for oxidizer and .5 psi for fuel in order to achieve a match with the input MRAPS derived vehicle mixture ratio of 1.602. The final interface pressure biases determined by this method were .5 psi and -.2 psi for oxidizer and fuel, respectively. Other parameters, specific impulse, thrust and total propellant consumed, remained essentially constant between the PAP run without the MRAPS results and the run including those results. It would be expected that PAP would match the MRAPS mixture ratio since other measurements included in the simulation are dependent on only total flowrate. It should be noted that the vehicle mixture ratio determined

[1] As a convention in this report, a negative bias indicates that measured data was reading less than its true value.

on the MRAPS derived value of mixture ratio. The 3 sigma limits on the MRAPS program results are ± .048 units (Reference 4).

The principal indicator of the accuracy of the postflight reconstruction is the matching of calculated and measured acceleration data. A measure of the quality of the match is given by the residual slope and intercept data as shown in Figure 2. These data represent the intercept, on the ordinate, and slope of a linear fit to the residual data. The closer both these numbers are to zero, the more accurate is the match. The acceleration match achieved with the LM-8 postflight reconstruction is excellent. The LM-8 flight reconstruction was by all indications an accurate simulation of actual flight performance.

The total propellant residuals at engine cutoff signal of the insertion burn from the results of the above data analysis were 201 lbm oxidizer and 125 lbm fuel. Based on these residual propellants, the remaining burn time capability of the ascent stage at APS first burn engine cutoff was approximately 28.5 seconds.

A vehicle damp weight reduction of 20 lbm was determined from the PAP reconstruction. The best estimate of total ascent stage weight at liftoff is 10,760 lbm.

Figures 2 through 9 show the principal performance parameters associated with the LM-8 postflight analysis. Four flight measurements were used as time varying input to the Propulsion Analysis Program. Two of these measurements, fuel and oxidizer interface pressure, were used as program drivers. The other two, acceleration and chamber pressure, were compared to calculated values by the program's minimum variance technique. The acceleration and chamber pressure measurements along with their residuals are presented in Figures 2 and 3, respectively. Figures 4 and 5 contain oxidizer and fuel interface pressure measurement data as they appeared after smoothing of the raw data, the curve fits of these data that were ultimately input to the Apollo Propulsion Analysis Program, and the residuals between the two data sets. Calculated steady-state values for the following parameters are shown in Figures 6-9: thrust, specific impulse, oxidizer flowrate and fuel flowrate.

Comparison with Preflight Performance Prediction

Predicted performance of the LM-8 APS is presented in Reference 8. The intention of the preflight performance prediction was to simulate APS performance under flight environmental conditions for the Mission H3 duty cycle. No attempt was made in the preflight prediction to simulate RCS operation.

Table 5 presents a summary of actual and predicted APS performance during the ascent burn. Measurement data compare quite closely with the reconstructed parameters. Engine specific impulse determined by the postflight reconstruction is slightly less than had been predicted but is still well within the 3-sigma limits of ± 3.5 seconds presented in Reference 8. Comparisons of predicted and reconstructed values for specific impulse, thrust, and mixture ratio are presented in Figure 10 along with related 3-sigma dispersions. The variations in flight specific impulse, thrust and mixture ratio were within their respective 3-sigma dispersions.

Engine Performance at Standard Interface Conditions

Expected APS engine flight performance was based on an engine characterization which utilized data obtained during engine and injector to be separated from variations induced by feed system, pressurization system, and propellant temperature variations, the acceptance test data are adjusted to a set of standard interface conditions, thereby providing a common basis for comparison. Standard interface conditions are as follows:

Oxidizer interface pressure, psia	170.
Fuel interface pressure, psia	170.
Oxidizer interface temperature, °F	70.
Fuel interface temperature, °F	70.
Oxidizer density, lbm/ft^3	90.21
Fuel density, lbm/ft^3	56.39
Thrust acceleration, lbf/lbm	1.
Throat area, in^2	16.47

Analysis results (at 13 seconds from ignition) for the ascent burn corrected to standard interface conditions and compared to acceptance test values are shown below:

	Acceptance Test Data	Flight Analysis Results	% Difference
Thrust, lbf	3502.	3495.	-.2
Specific Impulse, $\frac{lbf\text{-}sec}{lbm}$	310.3	309.8	-.2
Propellant Mixture Ratio	1.604	1.604	0

Reduction of engine performance to standard interface conditions and comparison with acceptance test values shows good agreement with the largest difference being in the engine specific impulse. All differences are within two standard deviations of acceptance test values.

It should be noted that due to the limited number of flight measurements available for use in determining APS propulsion system performance, it is not possible to independently determine engine and/or feed system resistance variations. As an example, given a system mixture ratio shift, it would not be possible to determine if the shift were attributable to the engine or feed system alone or was a result of the interaction of the two. It is apparent, therefore, that the adjustment of feed system data to standard engine inlet conditions could conceivably mask actual engine perturbations.

Helium Utilization

The helium storage tanks were loaded to a nominal 13.2 lbm. There was no indication of leakage from the helium bottles during the mission and calculated usage agrees well with analytical predictions.

Helium Bottle Pressure Prior to Ignition

During APS propellant tank pressurization a shift was noted in two of the helium bottle pressure measurements; GP0041 and GP0042. Following the firing of squib valves 1 and 2, GP0041 and GP0042 were reading 100-160 psi less than the other two helium bottle pressure measurements, GP0001 and GP0002. These pressure differences remained essentially constant up to and following ignition. Since there was no APS performance degradation and no indication, other than the measurement shift, of system abnormality, it was concluded that measurements GP0041 and GP0042 had been affected by the firing of the squib valves and that the magnitude of their readings was erroneous.

Helium Regulator Performance

No oscillations were noted in either helium regulator outlet pressure measurement. Oscillations of 6-19 psi have been noted in previous flight data. Also, oscillations of a similar nature and approximately twice that magnitude were noted during preflight checkout of the APS Class I secondary helium regulator.

The helium regulator outlet pressure measurements indicated pressures of approximately 1-2 psi less than the preflight predicted pressure of 184 psi. Interface pressure data reflect the reduced regulator outlet pressure. The helium regulator outlet pressure was within the Class I primary control band throughout the APS lunar liftoff burn.

Ullage Pressure Decay During Coast

Decay of the propellant tank ullage pressures is observed indirectly through the fuel and oxidizer interface pressures which at launch were 158 and 131 psia, respectively. At approximately 62 hours AET, these pressures had, as expected, decayed to 148 and 111 psia, respectively. This pressure drop is attributed to absorption of helium into the propellants. Pre-ignition pressurization of the propellant tank ullages was evidenced by the increase in both interface pressures to a value of approximately 186 psia at 141:27 hours AET.

Ullage Pressure Between APS First and Second Burn

During the lunar orbit following APS cutoff, both interface pressures quickly increased from their respective flow pressures to lock-up pressure, and then continued to increase by a total of about 16 psi on the oxidizer side and about 6 psi on the fuel side. Approximately twenty minutes after shutdown, with the interface pressures at 196 psi for oxidizer and 189 psi for fuel, loss of signal occurred as the vehicle went behind the moon. Predicted interface pressures for the period just prior to the second burn ignition were 195 psi for the oxidizer side and 190 psi for the fuel side. Second burn ignition occurred approximately eighteen minutes after loss of signal. At reacquisition of signal, interface pressures were 190 psi for oxidizer and 183 psi for fuel. The pressures were at approximately that same level at docking and rose to 197 psi and 184 psi, for oxidizer and fuel, respectively, prior to LM/CSM final separation.

APS propellant loads for the LM-8 Mission were 3218.2 lbm of oxidizer and 2007.0 lbm of fuel. Of these amounts 35.9 lbm of oxidizer and 16.0 lbm of fuel are considered to be unusable (except for depletion burns) or consumed during transient engine operation. The amounts of nominally deliverable propellants are, therefore, 3182.3 lbm and 1991.0 lbm for oxidizer and fuel, respectively. Propellant density samples taken at the time of loading showed an oxidizer density of 1.4812 gm/cc at 4°C and 14.7 psia and a fuel density of 0.8998 gm/cc at 25°C and 14.7 psia.

Since all RCS propellant usage was from the RCS tanks prior to lunar liftoff, the APS propellant loads at APS ignition were 3218.2 lbm of oxidizer and 2007.0 lbm of fuel. All RCS consumption during the ascent burn was through the APS/RCS interconnect. Total propellant usage from the APS tanks is presented in Table 3. The APS consumption during the lunar liftoff burn was 2970 lbm, oxidizer and 1860 lbm, fuel. Total RCS consumption during the APS first burn was 71 lbm. The TPI maneuver consumed an estimated 26 lbm of oxidizer and 16 lbm of fuel. A total of 175 lbm of oxidizer and 108 lbm of fuel remained onboard at APS second burn cutoff.

7. ENGINE TRANSIENT ANALYSIS

An analysis of the start and shutdown transients was performed primarily to determine the transient total impulse. Figures 11 and 12 are traces of engine chamber pressure, GP2010, during start and shutdown of the lunar liftoff burn, respectively. No data were available from the TPI burn.

The time from ignition signal to 90 percent steady-state thrust was 0.365 seconds, well within the specification limit for unprimed starts of 0.450 seconds. Total start transient impulse was 18 lbf-sec. The chamber pressure overshoot exceeded the upper limit of the measurement range (150 psia), however, there were no indications of rough combustion or other abnormal performance.

Total impulse from engine cutoff signal to 10 percent thrust was 372 lbf-sec. Time from cutoff signal to 10 percent thrust was 0.23 seconds which is within the revised specification limit of 0.500 seconds (Reference 9).

The LM-8 APS flight reconstruction showed the APS performance to be satisfactory. No malfunctions or anomalies with impact on future flights were noted.

Postflight analyses of LM-3, LM-4, LM-5 and LM-6 indicated that engine performance predictions might be somewhat conservative in that predicted specific impulse was less than the reconstructed value for all of these flights. The LM-8 postflight is an exception since the predicted average specific impulse is .5 seconds greater than the average reconstructed specific impulse at standard interface conditions. Reference 10 details a statistical study which includes the results of all APS postflight analyses. The results of this study show that there is no statistically significant specific impulse bias associated with APS postflight results. This conclusion will be verified for succeeding flights by expanding the data set to include results of subsequent analyses.

REFERENCES

1. NASA Document, MSC-04112, "Apollo 14 Mission Report," April, 1971.

2. North American Rockwell Corporation Document No. PAR 8114-4102, "Lunar Module Ascent Engine Performance Characterization," T. A. Clemmer, 10 July 1968.

3. Rocketdyne Engine Log Book, "Acceptance Test Data Package for Rocket Engine Assembly - Ascent LM - Part No. RS00058-001-00, Serial No. 0006," 3 February 1969.

4. Boeing Report D2-118346-1, "Computer Program Manual Ascent Propulsion Subsystem Mixture Ratio By the Center of Mass Method," 28 August 1970.

5. TRW Letter 71.6522-4-8, "Apollo 14 Postflight Mass Properties," 24 March 1971.

6. NASA Memorandum EP-23-10-69, from EP/2 Head Development Section to EP/2 Chief, Primary Propulsion Branch, "Propellant Densities (N_2O_4 and A-50)," 18 February 1969.

7. Spacecraft Operational Data Book, SNA-8-D-027(III), Rev. 2, Vol. III, Amendment 88,

8. TRW Report 17618-H054-R0-00, "Apollo Mission H3/LM-8/APS Preflight Performance Report," November 1970.

9. LTX-965-219, "P.O. 6-20900-C, LM Ascent Engine, LVC 275-005-051A," 2 July 1969.

10. TRW IOC, "Task 705-3, Subtask 4 Input," 1 September 1971.

Burn	Ignition Hr:Min:Sec AET	Engine Cutoff Hr:Min:Sec AET	Burn Time Seconds	Velocity (1) Change ft/sec
Lunar Liftoff	141:45:40.0	141:52:52:1	432.1	6066.1
TPI	142:30:51.1	142:30:54.7	3.6	88.5

(1) Reference 1

TABLE 2. LM-8/APS ENGINE AND FEED SYSTEM PHYSICAL CHARACTERISTICS

Engine[1]	
Engine No.	Rocketdyne S/N 0006C
Injector No.	Rocketdyne S/N 4097715
Initial Chamber Throat Area (in^2)	16.336[4]
Nozzle Exit Area (in^2)	749.073
Initial Expansion Ratio	45.854
Injector Resistance ($lbf\text{-}sec^2/lbm\text{-}ft^5$)@ time zero and 70°F	
Oxidizer	12586.
Fuel	20342.
Feed System	
Total Volume (Pressurized, Check Valves to engine interface)(ft^3)[2]	
Oxidizer	36.89
Fuel	37.00
Resistance, Tank Bottom to Engine Interface ($lbf\text{-}sec^2/lbm\text{-}ft^5$) at 70°F[3]	
Oxidizer	2580.48
Fuel	4102.56

(1) Rocketdyne Log Book, "Acceptance Test Data Package for Rocket Engine Assembly-Ascent LM-Part No. RS000580-001-00, Serial No. 0006," 3 February 1969.

(2) NASA Memorandum EP23-46-69, "Propellant Load Parameters for the DPS and APS of LM-5 Through LM-9 and the Estimated Parameters for LM-10 and Subsequent," from EP/Chief, Propulsion and Power Division to PD/Chief, Systems Engineering Division.

(3) GAC Memorandum LMO-271-844, "A/S Hydraulic Resistance LM 7, 8, 9," W. Salter, 6 December 1969.

(4) The initial throat area determined from postflight reconstruction was 16.45 in^2.

TABLE 3. PROPELLANT CONSUMPTION FROM APS TANKS

	Oxidizer	Fuel
Loaded - lbm	3218.2	2007.0
Consumed During Lunar Liftoff Burn - lbm		
APS	2970.1	1859.9
RCS	47.0	23.5
Total Propellant Remaining - lbm	201.1	123.6
Consumed During TPI Burn - lbm		
APS	26.2	15.8
Total Propellant Remaining - lbm	174.9	107.8

Measurement Number	Description	Range	Sample Rate Sample/sec
GP2010P	Pressure, Thrust Chamber	0-150 psia	200
GP1503P	Pressure, Engine Oxidizer Interface	0-250 psia	1
GP1501P	Pressure, Engine Fuel Interface	0-250 psia	1
GP0025P	Pressure, Regulator Outlet Manifold	0-300 psia	1
GP0018P	Pressure, Regulator Outlet Manifold	0-300 psia	1
GP1218T	Temperature, Oxidizer Tank Bulk	20-130°F	1
GP0718T	Temperature, Fuel Tank Bulk	20-130°F	1
GH1260X	Ascent Engine On/Off	Off-on	50
GP0001P	Pressure, Helium Supply Tank No. 1	0-4000	1
GP0002P	Pressure, Helium Supply Tank No. 2	0-4000	1
GP0041P	Pressure, Helium Supply Tank No. 1	0-4000	10
GP0042P	Pressure, Helium Supply Tank No. 2	0-4000	10
CG0001X*	PGNS Downlink Data	Digital Code	50

* Acceleration determined from PIPA data

TABLE 5. LM-8 APS STEADY-STATE PERFORMANCE

PARAMETER	30 Sec After Ignition			200 Sec After Ignition			400 Sec After Ignition		
	Pred. (a)	Reconstructed (b)	Measured (c)	Pred. (a)	Reconstructed (b)	Measured (c)	Pred. (a)	Reconstructed (b)	Measured (b)
Regulator Outlet Pressure (psia)	184.	---	183.0	184.	---	182.5	184.	---	181.5
Oxidizer Bulk Temperature °F	70.0	---	69.9	69.7	---	69.5	69.0	---	69.4
Fuel Bulk Temperature °F	70.0	---	69.5	69.9	---	69.5	69.8	---	69.0
Oxidizer Interface Pressure, psia	170.9	168.6	169.1	170.7	168.4	168.7	170.0	167.3	167.8
Fuel Interface Pressure, psia	170.4	169.0	169.0	170.2	168.5	168.3	169.6	167.3	166.7
Engine Chamber Pressure, psia	124.1	122.1	121.4	124.4	122.3	121.8	123.7	120.6	120.5
Mixture Ratio	1.602	1.598	---	1.597	1.599	---	1.594	1.597	---
Thrust, lbf	3516.	3479.	---	3489.	3457.	---	3486.	3458.	---
Specific Impulse, sec	310.1	309.9	---	310.2	309.9	---	309.6	309.0	---

(a) Preflight prediction based on acceptance test data and assuming nominal system performance.
(b) Reconstruction minimum variance technique.
(c) Smoothed flight data without biases determined by postflight analysis.

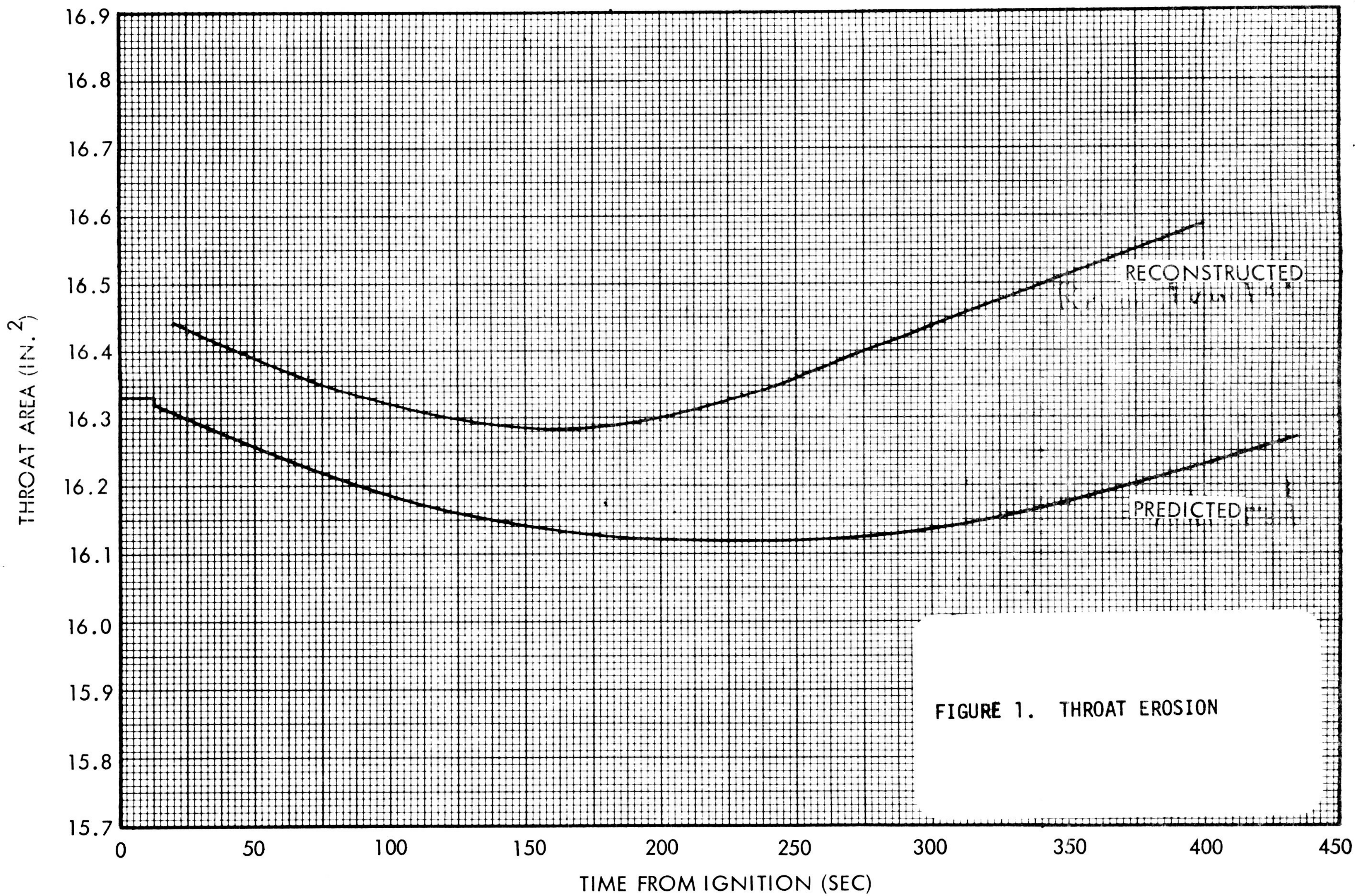


FIGURE 1. THROAT EROSION

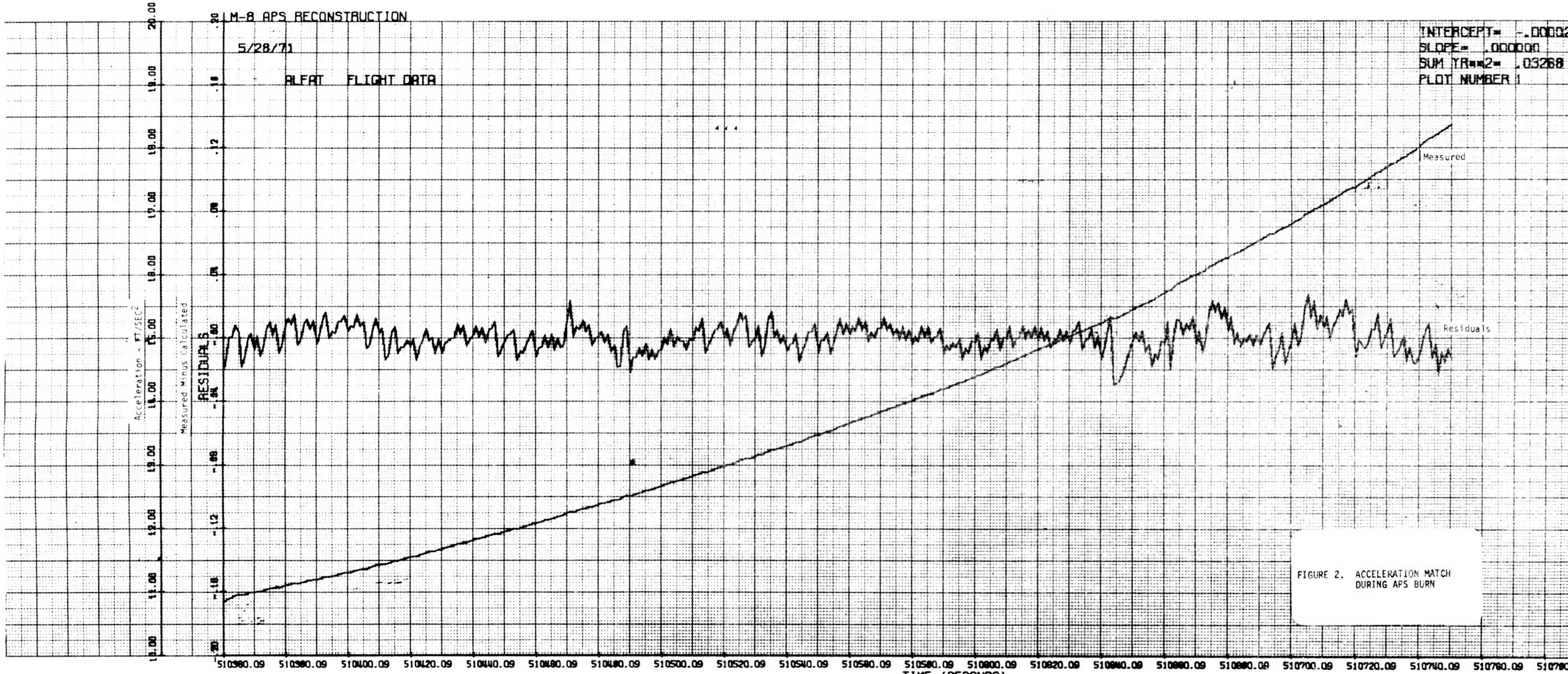


FIGURE 2. ACCELERATION MATCH DURING APS BURN

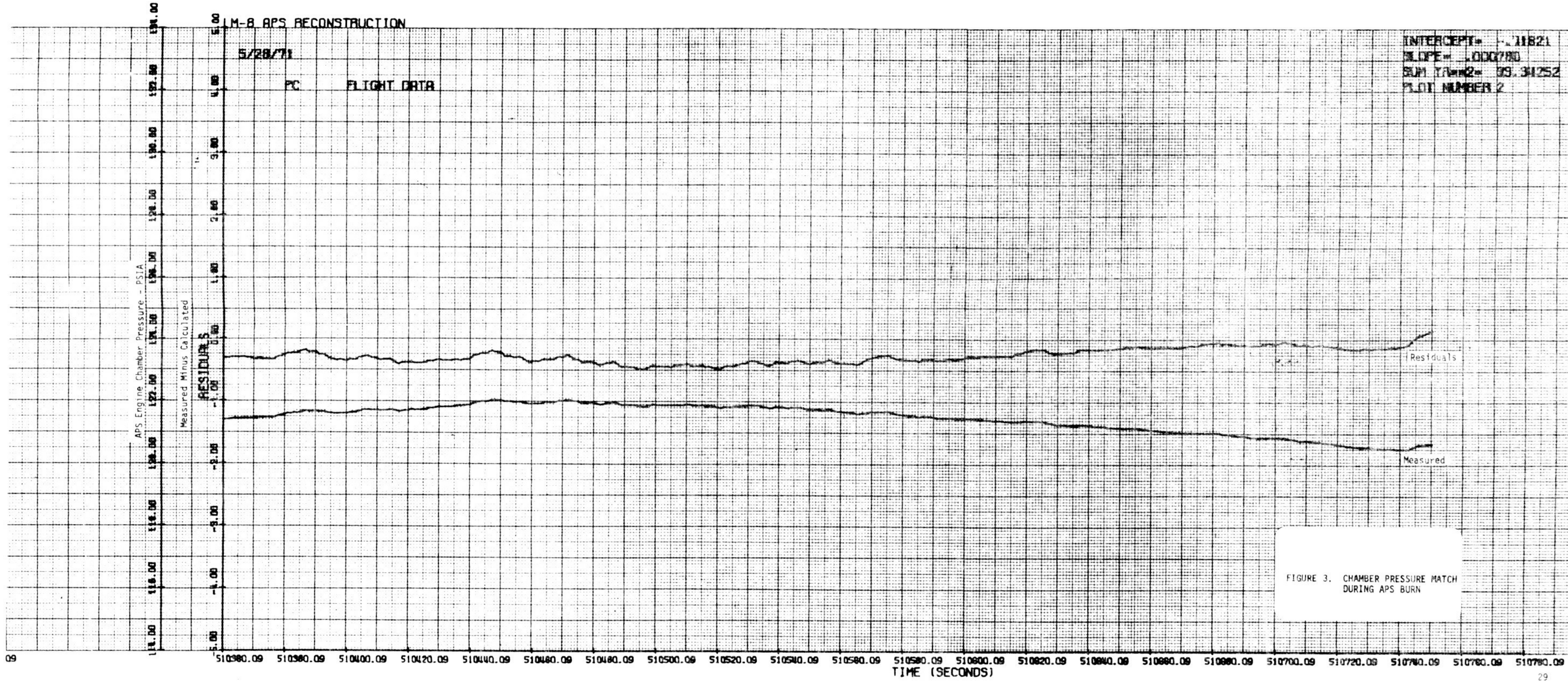


FIGURE 3. CHAMBER PRESSURE MATCH DURING APS BURN

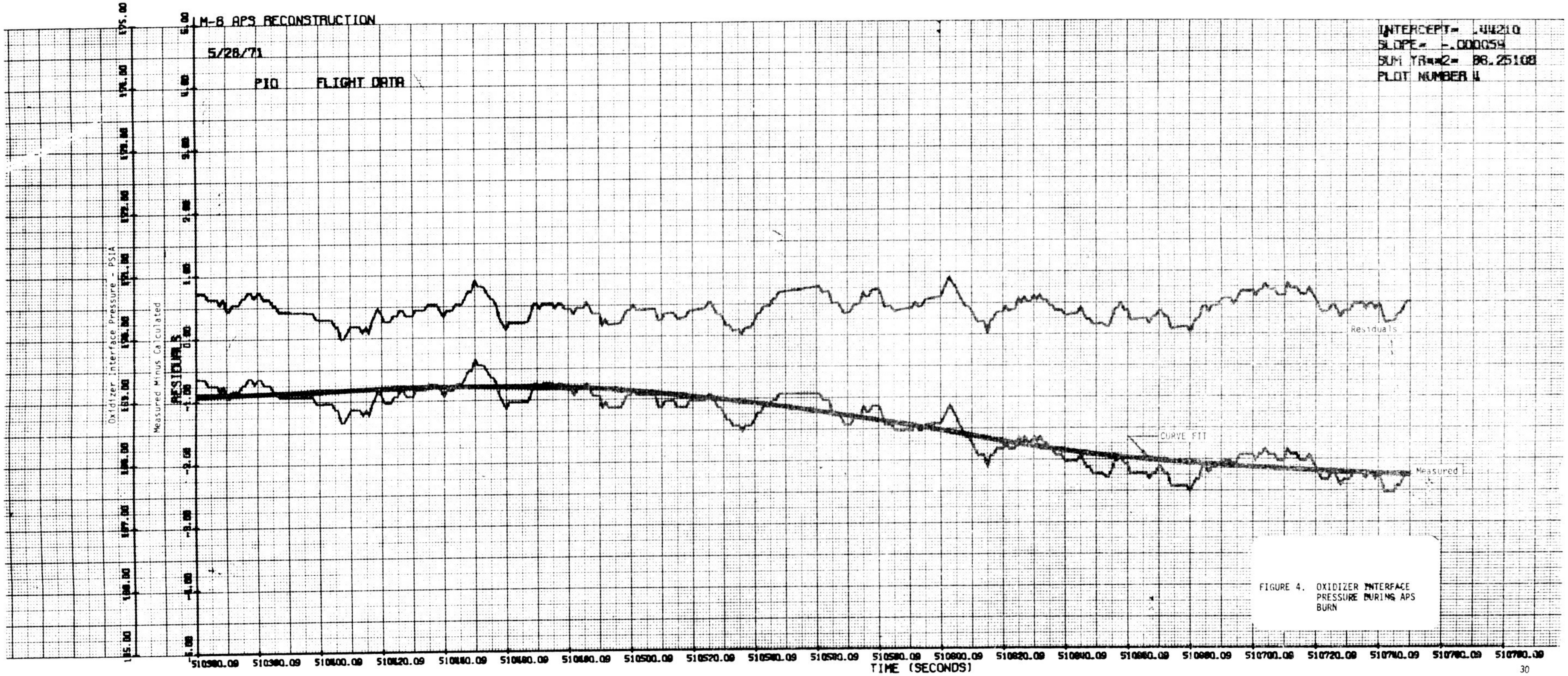


FIGURE 4. OXIDIZER INTERFACE PRESSURE DURING APS BURN

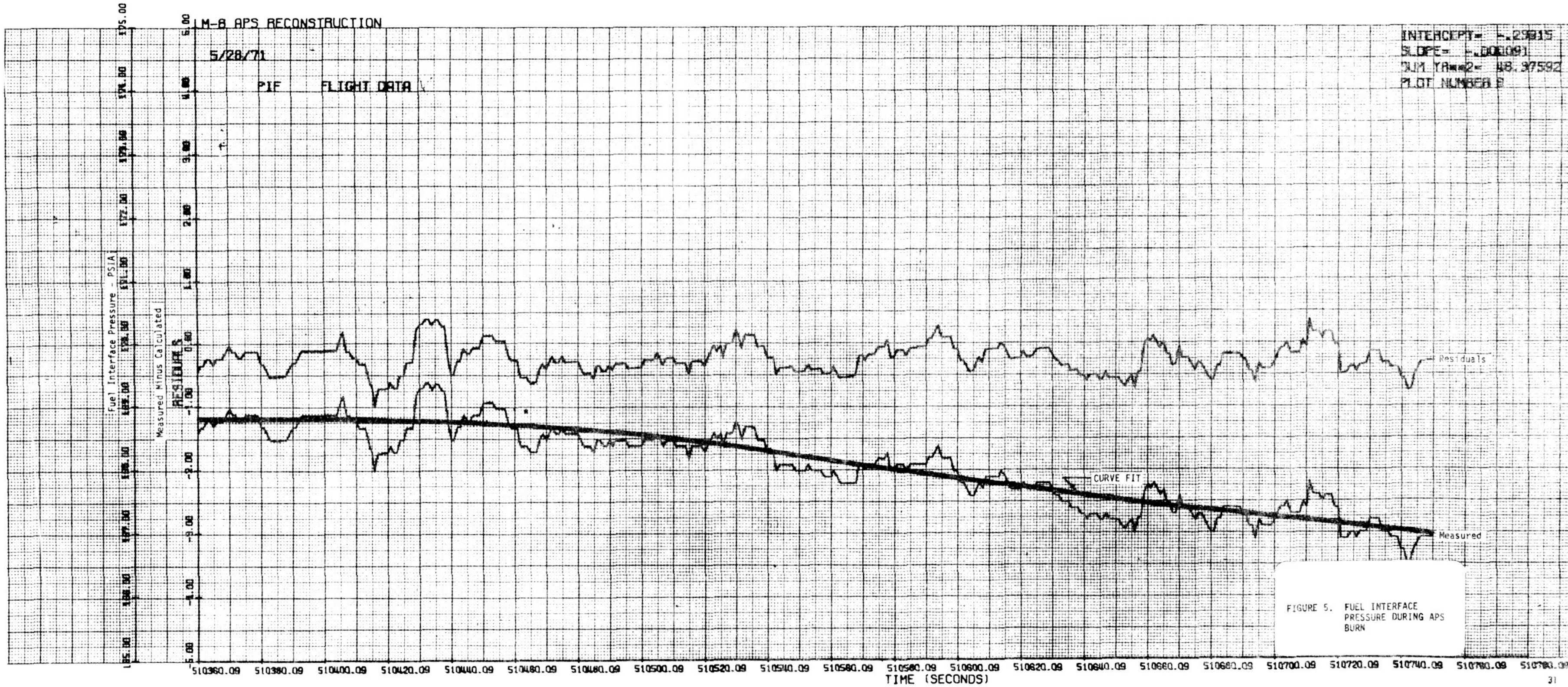


FIGURE 5. FUEL INTERFACE PRESSURE DURING APS BURN

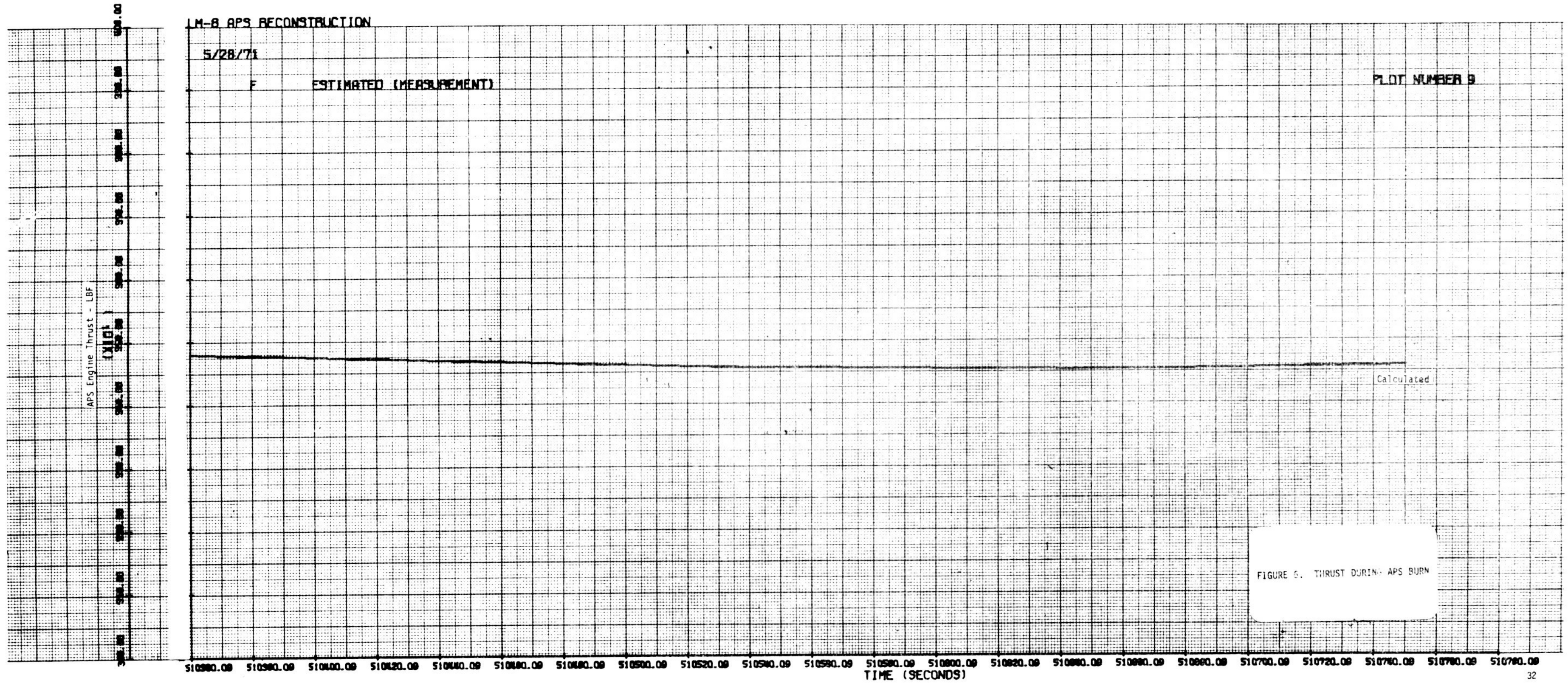


FIGURE 5. THRUST DURING APS BURN

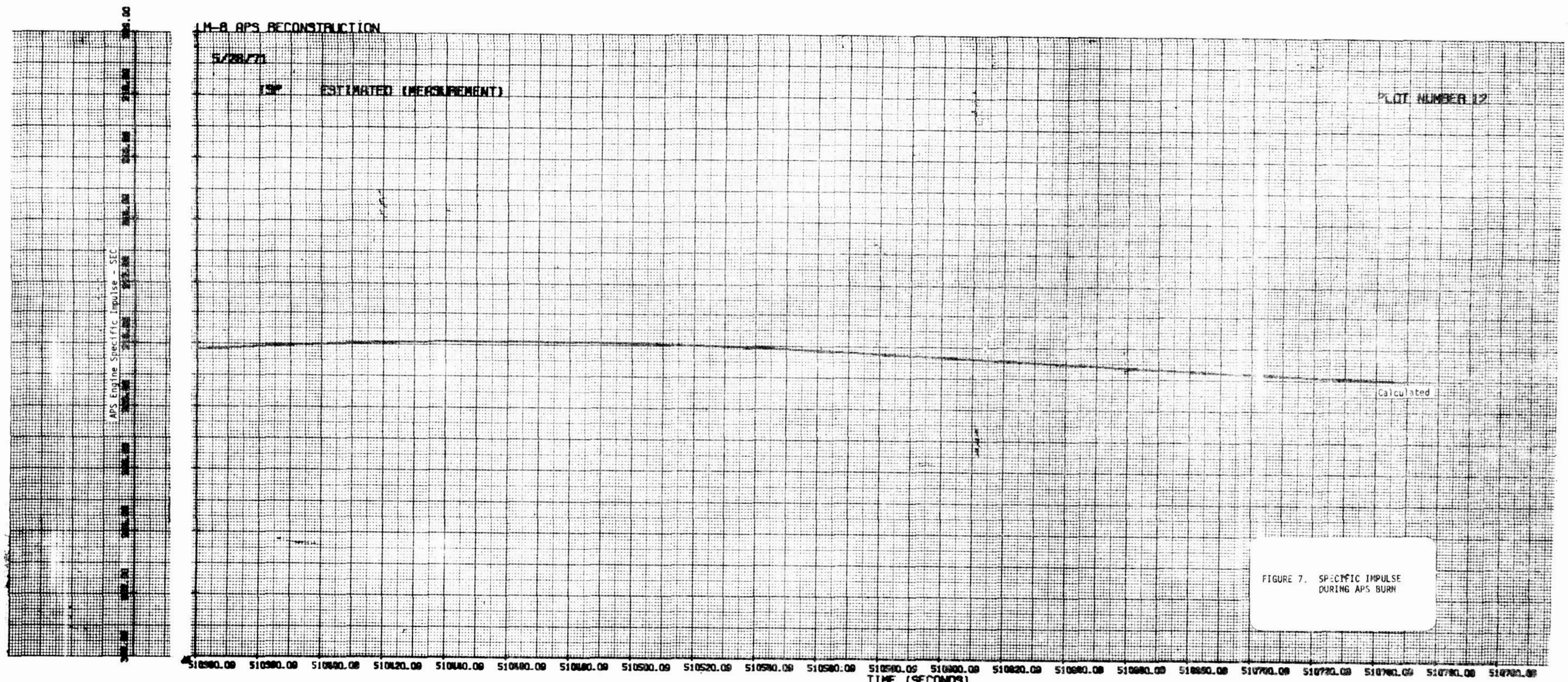


FIGURE 7. SPECIFIC IMPULSE DURING APS BURN

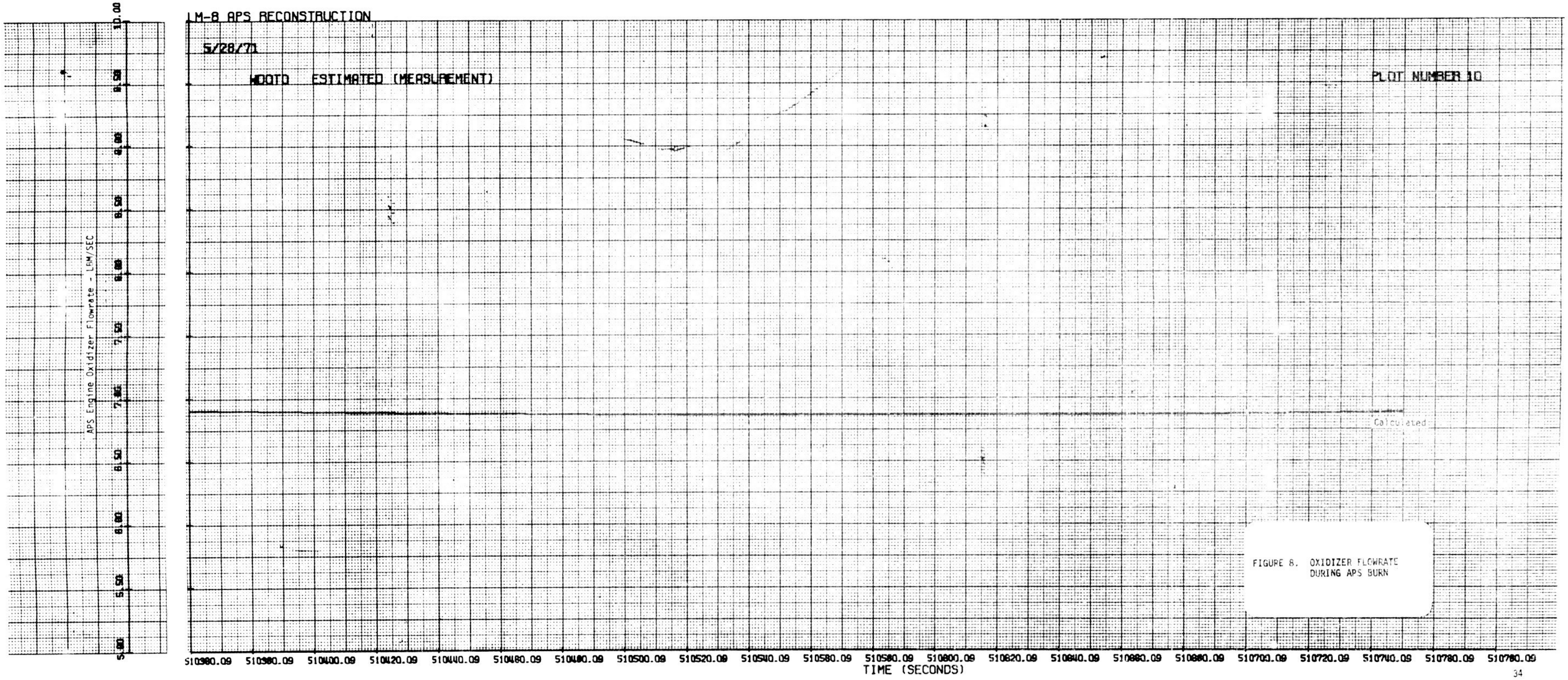


FIGURE 8. OXIDIZER FLOWRATE DURING APS BURN

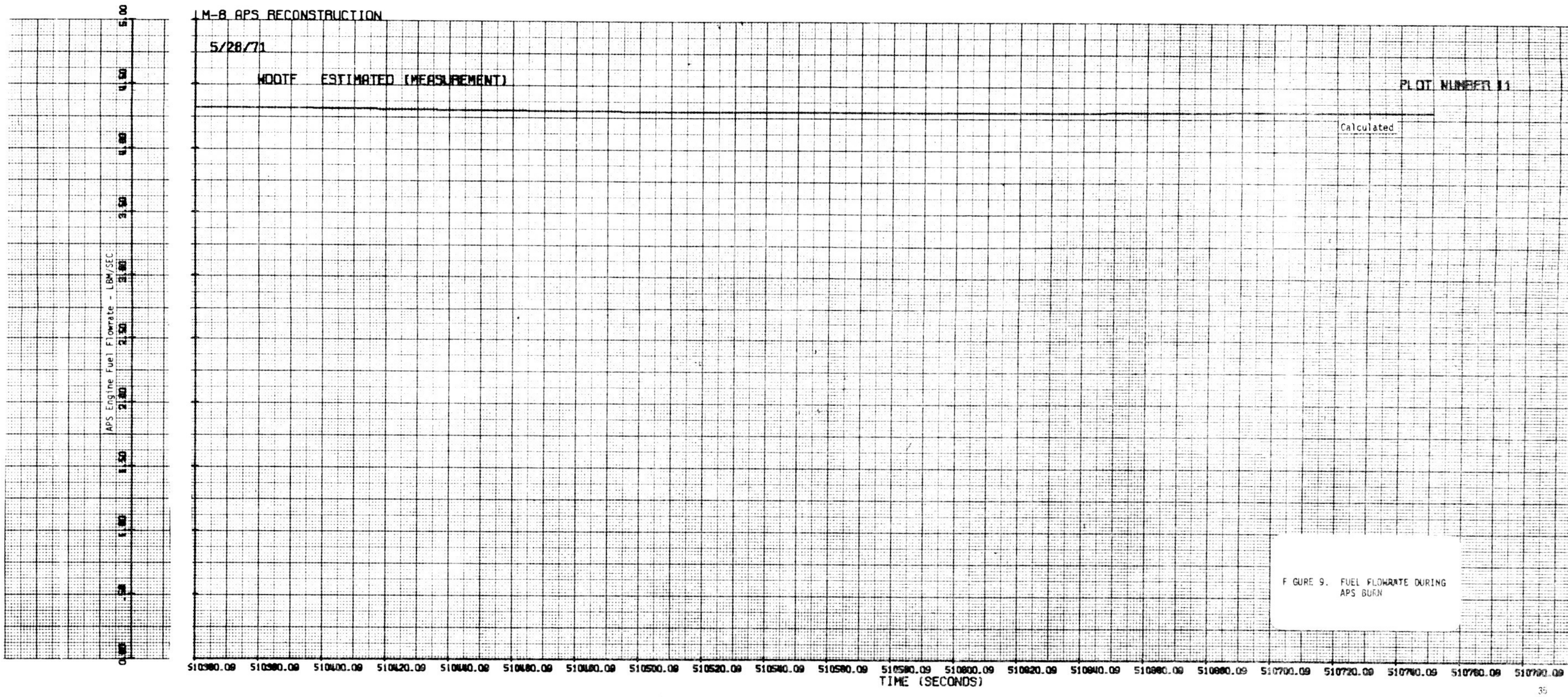


FIGURE 9. FUEL FLOWRATE DURING APS BURN

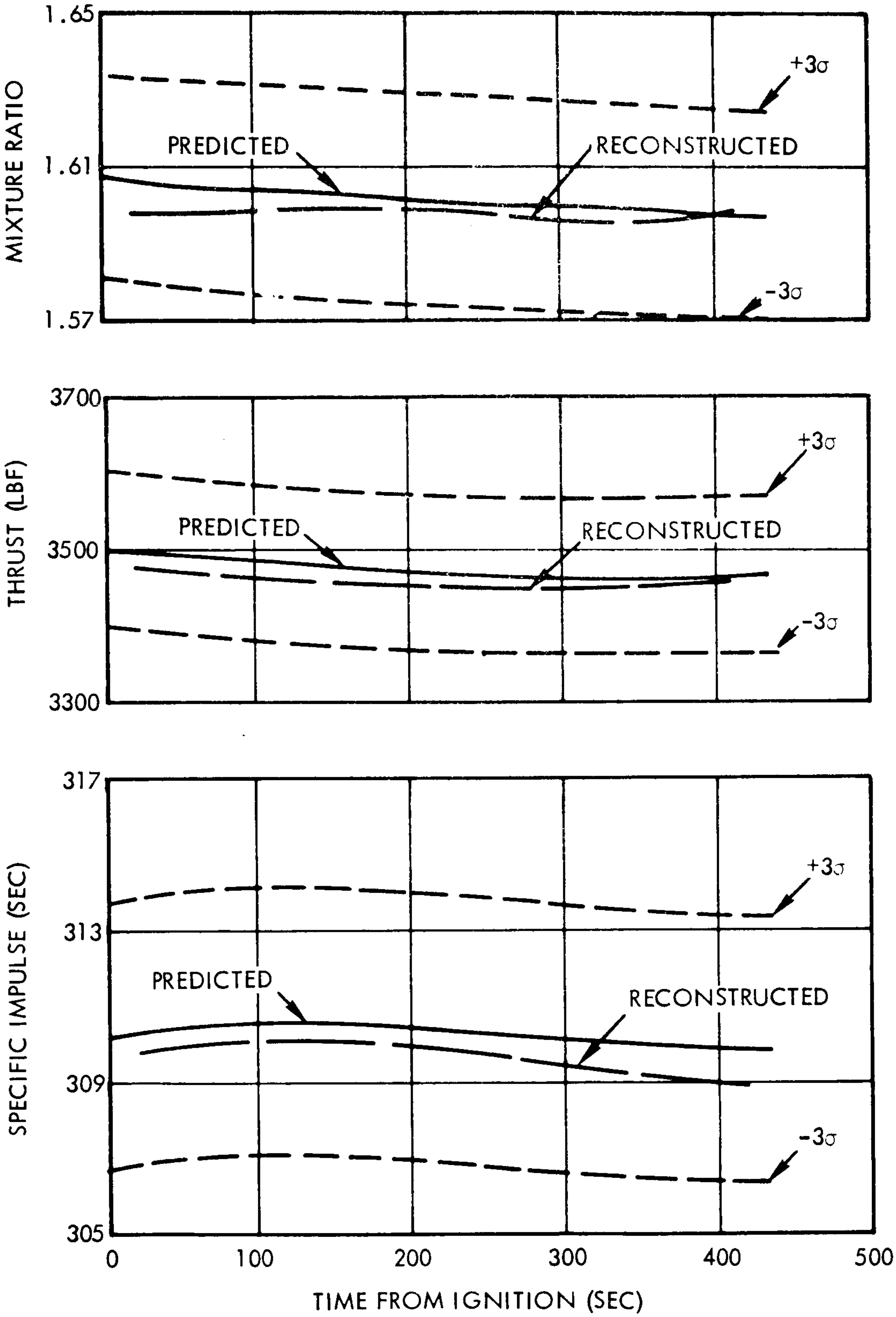


FIGURE 10

COMPARISON OF PREDICTED AND RECONSTRUCTED PERFORMANCE

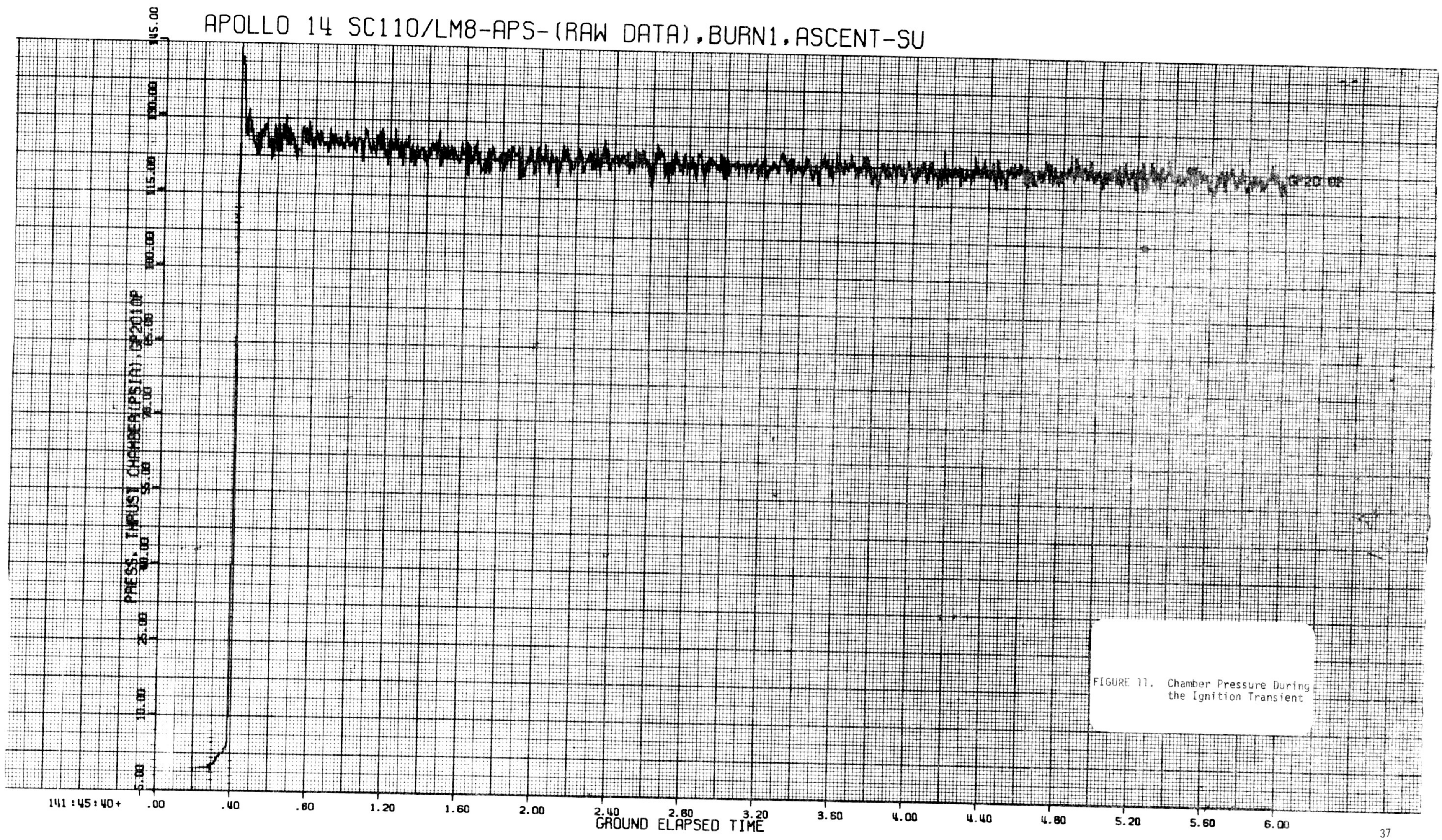


FIGURE 11. Chamber Pressure During the Ignition Transient

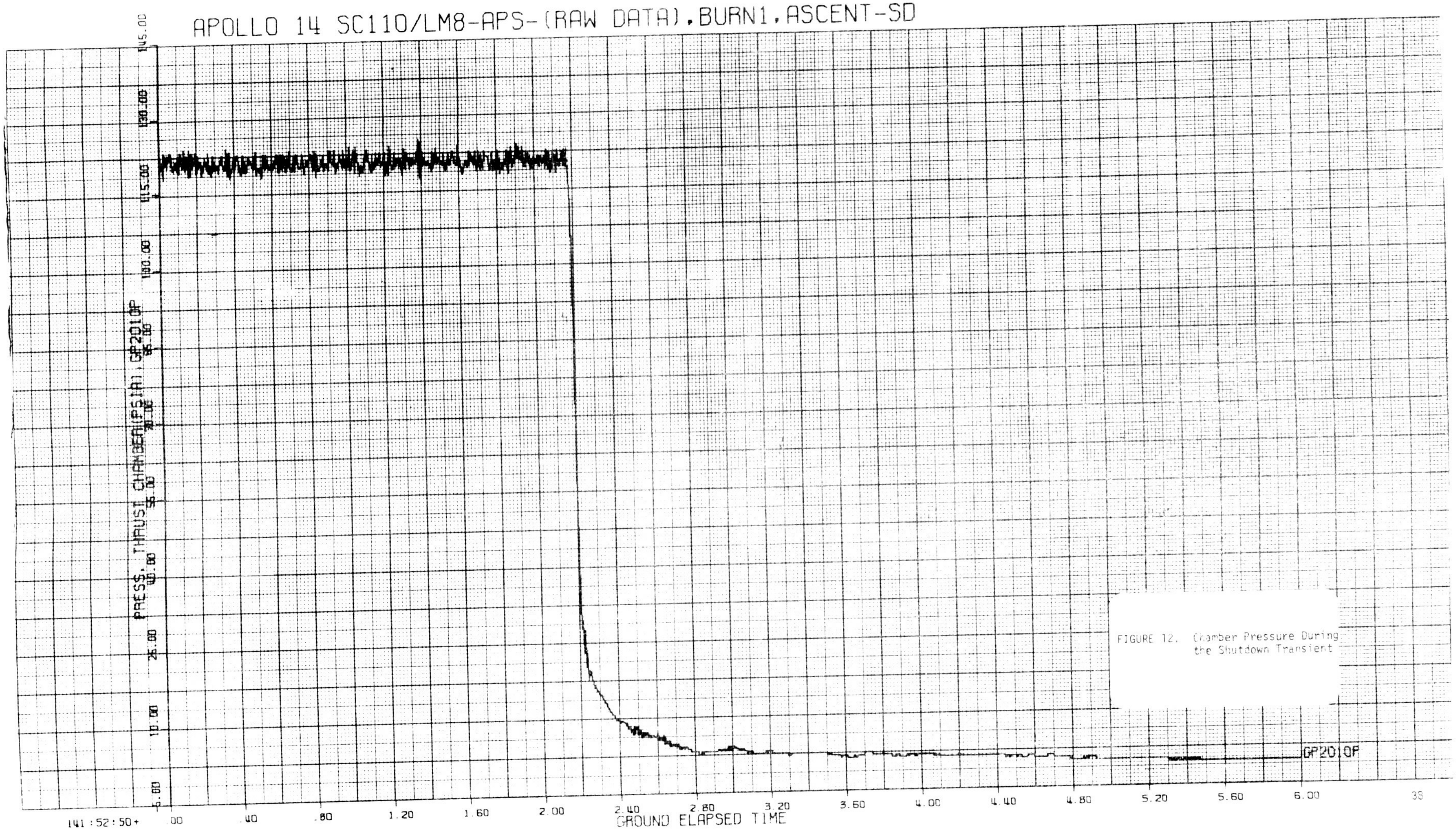


FIGURE 12. Chamber Pressure During the Shutdown Transient

APPENDIX

FLIGHT DATA

APOLLO 14 SC110/LM8-APS-(RAW DATA).BURN1.ASCENT

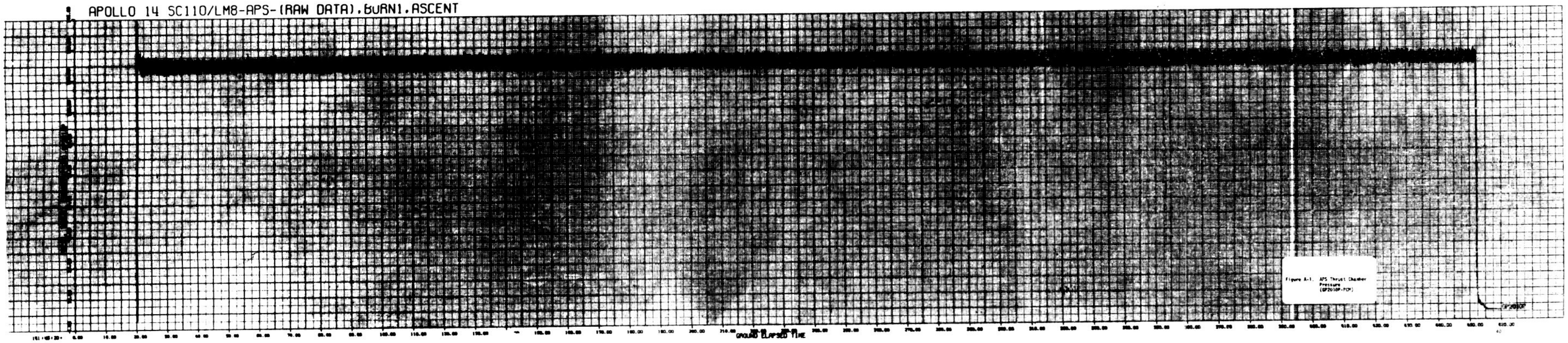


Figure A-1. APS Thrust Chamber Pressure (GP2010P-PCM)

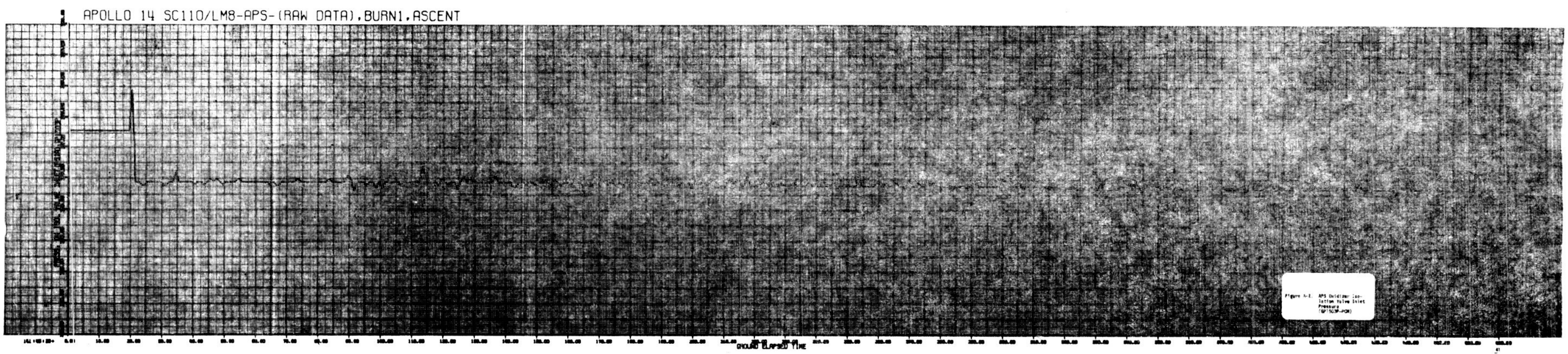
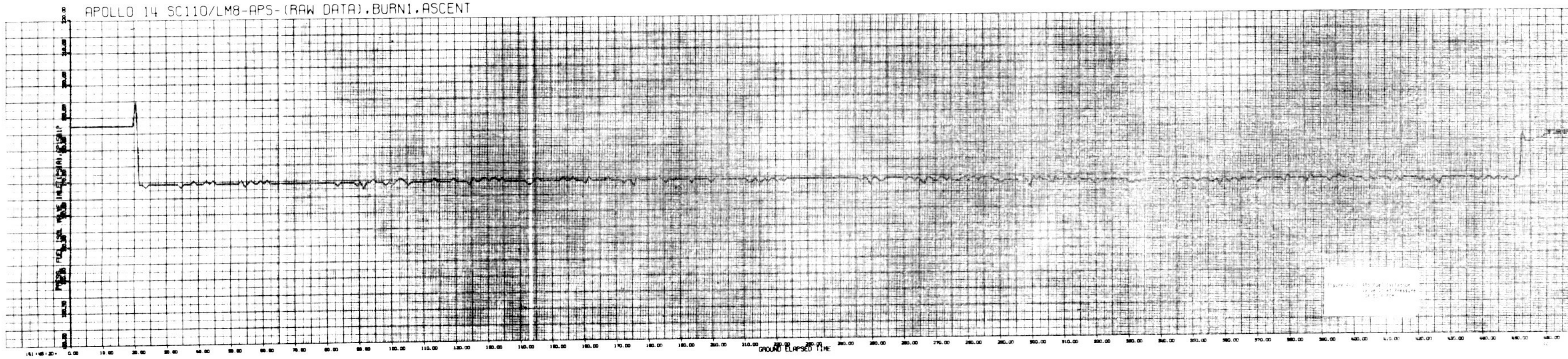


Figure A-2. APS Oxidizer Isolation Valve Inlet Pressure (GP1503P-PCM)

APOLLO 14 SC110/LM8-APS-(RAW DATA),BURN1,ASCENT



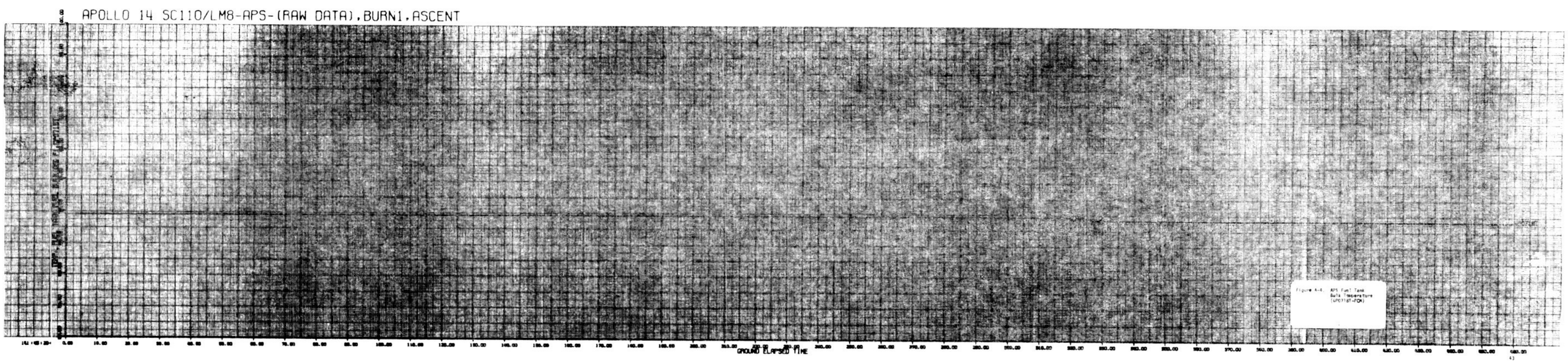
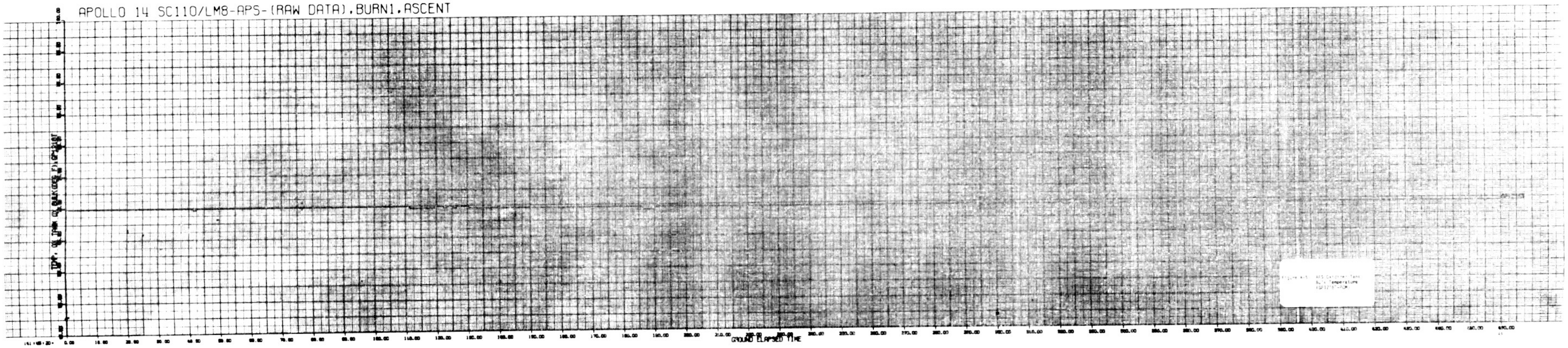


Figure A-4. APS Fuel Tank Bulk Temperature (GP0718T-PCM)

APOLLO 14 SC110/LM8-APS-(RAW DATA),BURN1,ASCENT



APOLLO 14 SC110/LM8-APS-(RAW DATA).BURN1.ASCENT

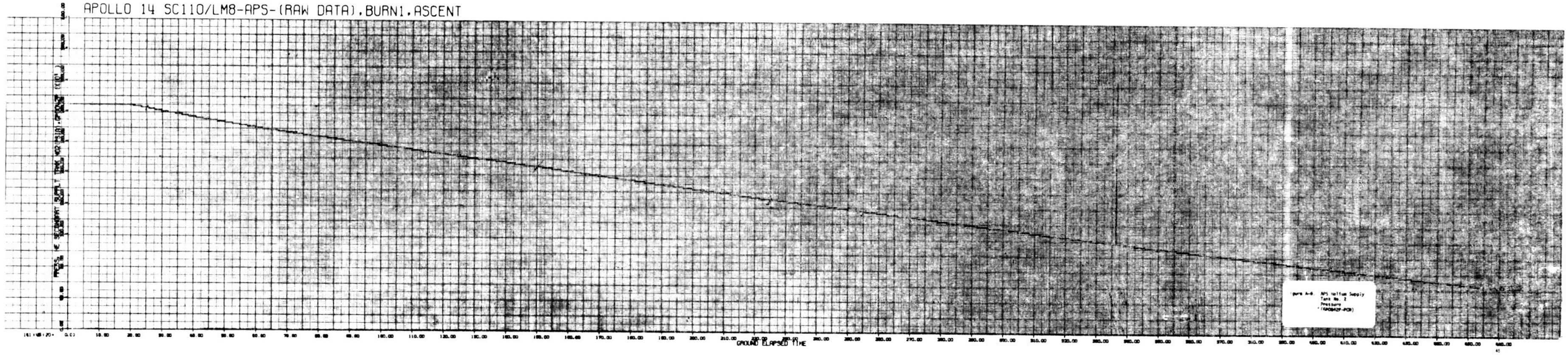


Figure A-6. APS Helium Supply Tank No. 2 Pressure (GP0842P-PCM)

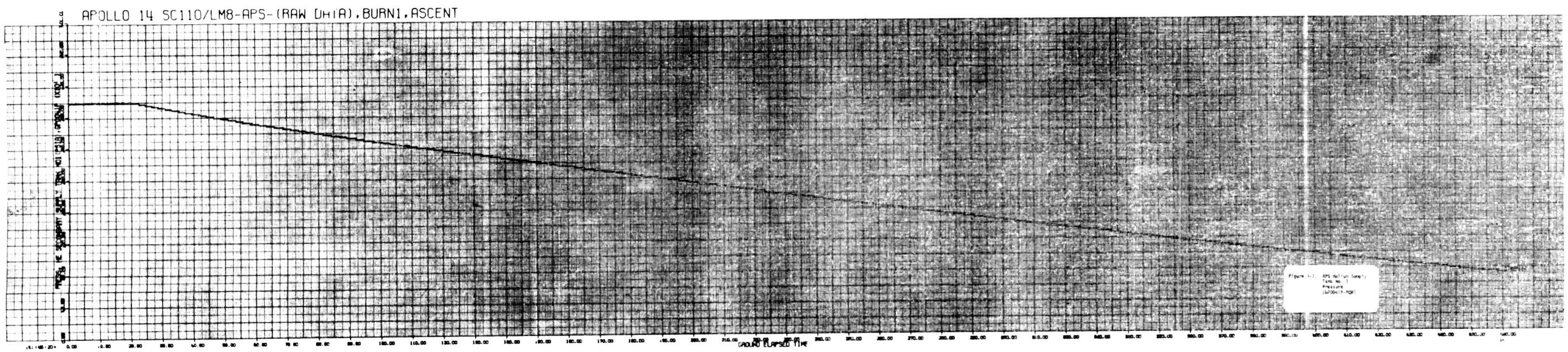


Figure [illegible]. APS Helium Supply Tank No. 1 Pressure (GP0001P-MQR)

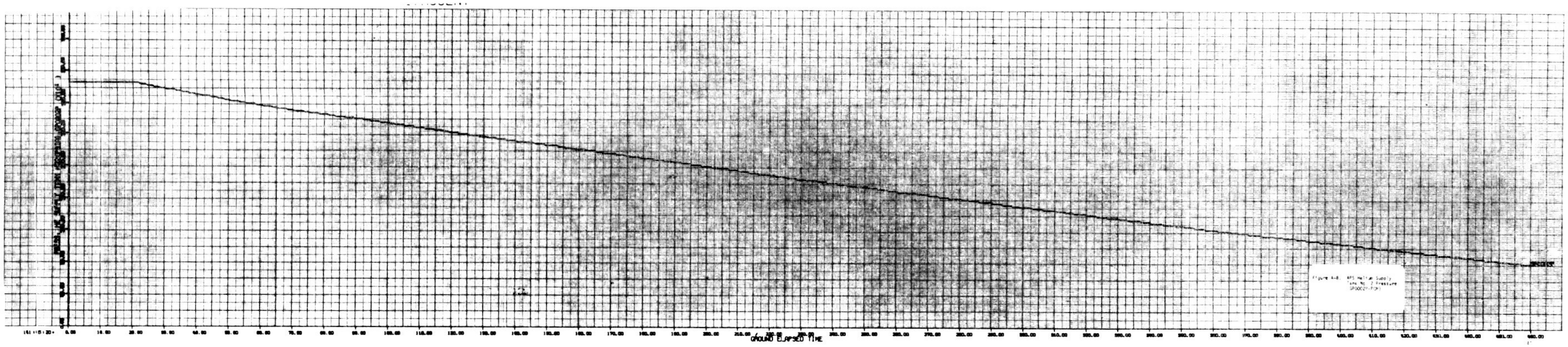
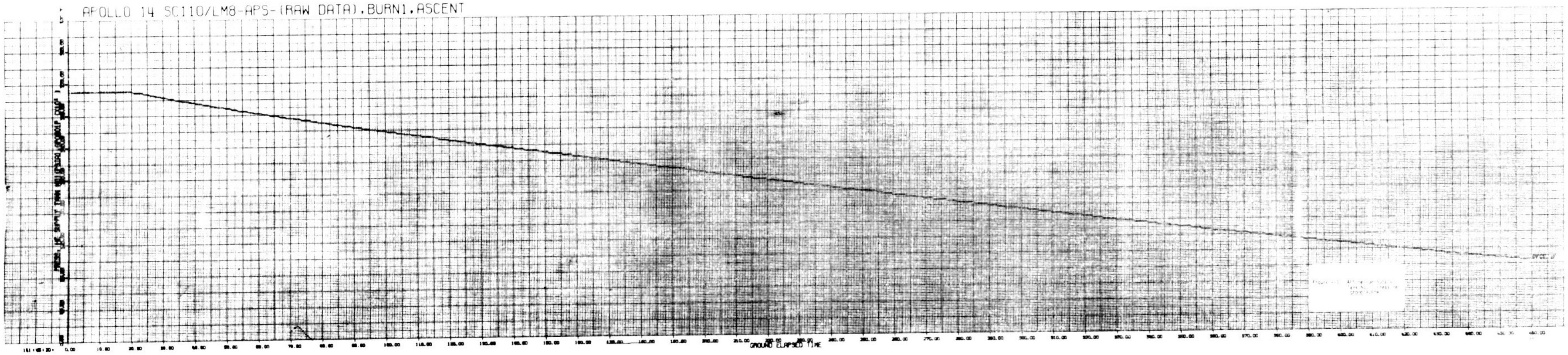


Figure A-8. APS Helium Supply Tank No. 2 Pressure GP0002P-P(M)

APOLLO 14 SC110/LM8-APS-(RAW DATA),BURN1,ASCENT



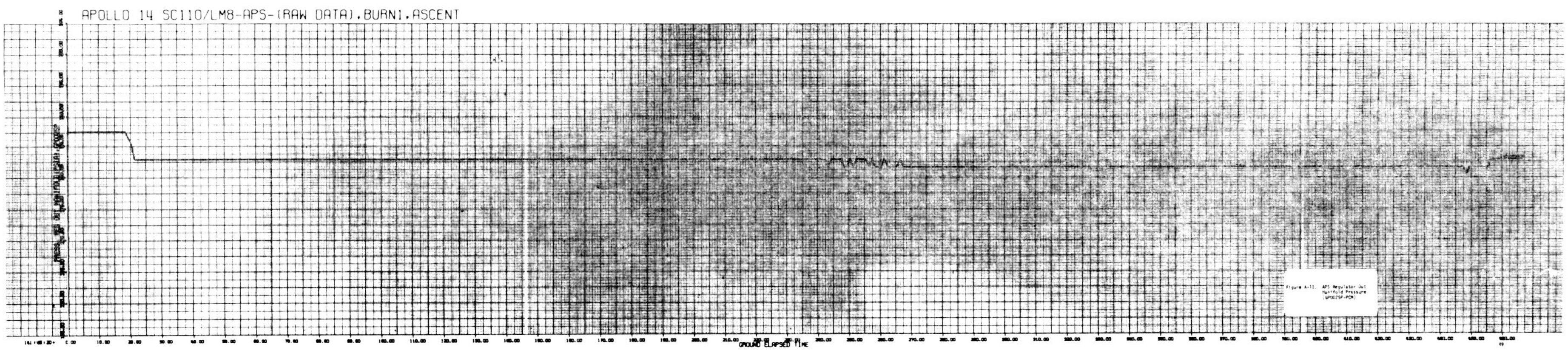
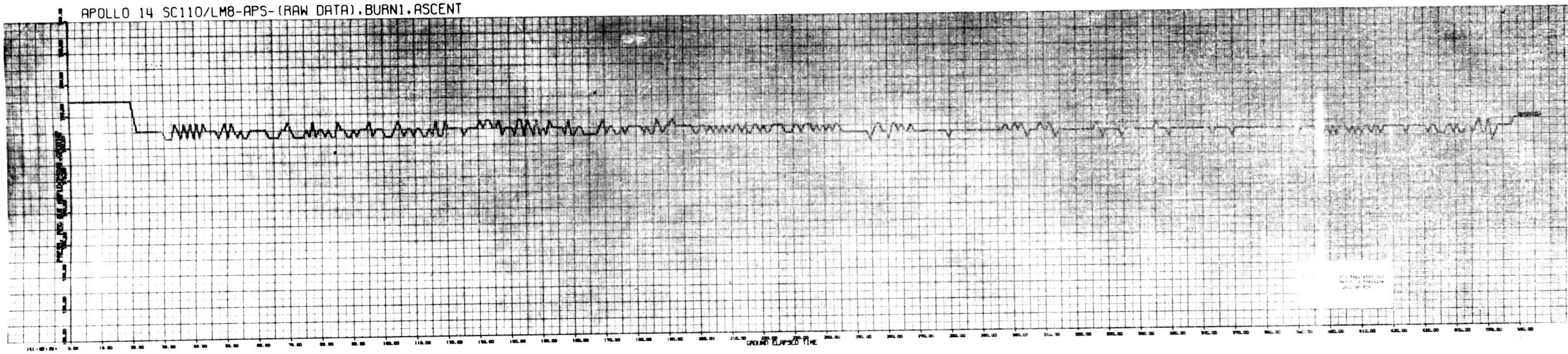


Figure A-10. APS Regulator Out Manifold Pressure (GP0025P-PCM)

APOLLO 14 SC110/LM8-APS-(RAW DATA).BURN1.ASCENT



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